Copy No. 4

~~CONFIDENTIAL~~

RM No. A7A02

H L Dryden per NACA release form #16 - 11/20/49 ac 4/16/53

APR 17 1947

NACA

RESEARCH MEMORANDUM

THE INTERACTION OF BOUNDARY LAYER AND COMPRESSION SHOCK
AND ITS EFFECT UPON AIRFOIL PRESSURE DISTRIBUTIONS

By

H. Julian Allen, Max. A. Heaslet,
and Gerald E. Nitzberg

Ames Aeronautical Laboratory
Moffett Field, Calif.

NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

WASHINGTON
April 10, 1947

~~CONFIDENTIAL~~

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

THE INTERACTION OF BOUNDARY LAYER AND COMPRESSION SHOCK AND ITS EFFECT UPON AIRFOIL PRESSURE DISTRIBUTIONS

By H. Julian Allen, Max. A. Heaslet,
and Gerald E. Nitzberg

SUMMARY

The mechanism of the interaction of compression shock with boundary layer is investigated. It is shown that the apparently shockless pressure distributions observed experimentally at supercritical Mach numbers can be accounted for by a marked thickening of the boundary layer for some distance ahead of the shock wave. Pressure distributions with abrupt pressure recovery from large local supersonic Mach numbers to a local Mach number of about unity can be accounted for by a thickening of the boundary layer over a shorter chordwise extent.

At the test Reynolds number of about 1,500,000, the presence of an aerodynamically clean or dirty surface does not materially influence the drag at high supercritical speeds.

INTRODUCTION

During calibration of the Ames 1- by $3\frac{1}{2}$-foot high-speed wind tunnel, a study of wall interference by the image method was undertaken to determine the validity of theoretical

[redacted]

interference factors. It was found in the course of the tests that pressure-distribution measurements at supercritical air speeds could not always be repeated. This phenomenon was particularly evident at certain angles of attack.

The fact that the phenomenon appeared only at supercritical speeds suggested that it was in some way brought about by the interaction between compression shock and the airfoil boundary layer. Such interaction is possible since a stationary shock wave, which only exists within a fluid where the velocity is above sonic, cannot penetrate to the surface of the airfoil where the velocity must be zero and, in consequence, the pressure rise across the shock wave must effect a tendency to a reversed flow within the boundary layer.

An investigation was undertaken to determine the cause of the phenomenon with a view to clarifying the present understanding of compression shock and the interrelation of shock and boundary layer.

EXPERIMENTAL INVESTIGATION

The Ames 1- by $3\frac{1}{2}$-foot high-speed wind tunnel, which was used in these tests, is a low-turbulence wind tunnel with sufficient power to obtain choked flow under all test conditions. The airfoil models used in the present tests were of NACA 4412 airfoil section and spanned the 1-foot dimension of the tunnel. There was no end leakage. One model, described in reference 1, which had a 5-inch chord, was equipped with 54 pressure orifices and was used to determine the distribution of pressure over the airfoil section. The other model had a 6-inch chord and was used to obtain schlieren photographs of the flow.

Of the airfoil pressure distributions for the wall-interference study, several obtained at a Mach number of 0.75 and an angle of attack of 4° are of major interest for the present study. The Reynolds number for this test was about 1.5 millions. The pressure distribution for the clean airfoil under these conditions is shown in figure 1.

It was noted during the tests that, after prolonged operation of the tunnel, the pressure distribution changed from that shown; and it was found that this change was brought about by an accumulation of dirt and a pitting due to gritty particles striking the surface. Polishing of the airfoil surface permitted the original pressure distribution on the airfoil to be reattained. This result suggested that the change in pressure distribution was brought about by a forward movement of the point of transition from laminar to turbulent flow. Accordingly, the pressures were measured with a thin spanwise strip of coarse carborundum (No. 180) fixed at the 6-percent-chord station on the upper surface of the airfoil. An even more pronounced change in the pressure distribution of the kind previously found with the dirty surface was observed. This distribution is shown in figure 1.

The distribution of pressures obtained with the clean airfoil surface is not compatible with the usual theory of supercritical flow since there is no sharp drop in pressure coefficient to indicate the presence of a shock. In order to study this anomaly by the schlieren method, an NACA 4412 airfoil model was mounted for test on circular glass side-wall disks. The attachment to each disk was effected by two steel pins, at the 10- and 70-percent-chord stations. In order to seal the end of the airfoil to the disks, thin rubber gaskets were used. Typical schlieren photographs obtained with this apparatus are shown in figure 2. It will be noted that, unfortunately, the mounting pins and edge of the gaskets are visible but they can readily be distinguished. In several instances, visual examination of the schlieren screen showed the shock wave to be blurred, indicating chord-wise motion of the wave. Such an unsteady flow could, of course, produce a pressure distribution such as that shown in figure 1 for the clean-surface model since the liquid manometer used for these tests would not respond to rapid changes in pressure. In many instances, however, although an apparently shockless pressure distribution was obtained, the schlieren screen indicated that a compression shock was present and stationary with time. This observation prompted an analysis of the experimental results obtained.

ANALYTICAL INVESTIGATION

The experimental pressure distribution obtained with a clean surface, shown in figure 3, is akin to the corresponding distribution of figure 1 in that, for the upper surface in both instances, although the Mach number is well above critical, evidence of shock is apparently absent although the schlieren photograph of figure 4(a), corresponding to the pressure distribution of figure 3, shows shocked flow. Such a pressure recovery, it was considered, could be produced by interaction of the shock wave and boundary layer.

In order to explain this, it is necessary to consider the general characteristics of the supersonic portions of pressure distributions at supercritical speeds in the absence of local boundary-layer effects. Prandtl and Meyer have shown that, when a semi-infinite uniform stream at a Mach number of unity is deflected around a convex surface, the local Mach number attained at any point is supersonic with magnitude a function only of the total angle through which the stream has turned. This theory is modified empirically in reference 2 for the fact that the supersonic region of flow over airfoils at supercritical speeds is of only limited extent. It is found that the local Mach number is still directly related to the total angle turned through by the surface from the sonic point to the point under consideration. However, the Mach number rise for a given angular deflection is only from one-third to two-thirds that predicted by Prandtl and Meyer, the precise value being determined by the specific configuration of airfoil shape, angle of attack, and free-stream Mach number.

It is known from studies of boundary-layer effects at low speeds that the presence of a boundary layer is equivalent to a proportionate local thickening of the airfoil. There is no reason to believe that this effect will be different at high speeds. It follows that, since supersonic local Mach numbers, and therefore local pressure coefficients, are directly affected by local surface contours, any change in boundary-layer thickness which markedly changes the effective local curvature of the airfoil surface will result in a marked local change in the airfoil pressure distribution. There is an abrupt pressure increase across a shock wave which can be transmitted forward only through the boundary layer. This pressure rise produces an adverse pressure gradient in the boundary layer which tends to thicken and, in some cases, actually to separate the boundary layer. A marked thickening of the boundary layer ahead of the shock wave may so modify the effective curvature of the airfoil surface that the original convexity, with the resultant falling pressure, in front of the wave can be changed to concavity with a resultant rising pressure in front of the wave. In other words, the effect of the thickening of the boundary layer can be such as to decrease the local Mach numbers outside the boundary layer as the shock wave is approached. The pressure rise across the shock wave is thereby diminished, resulting in a so-called "softened" shock wave. Donaldson, in reference 3, found that the concept of the softened compression shock led to the successful prediction of the position of the terminal shock wave in supersonic nozzles. His prediction was based upon the assumption that this softening was so complete as to make the Mach numbers immediately ahead of and behind the terminal shock wave essentially unity. If the supposition is made that the pressure distribution can be related to the effective surface curvature and if it is assumed, after Donaldson, that the terminal shock wave occurs when the surface pressure is that corresponding to a Mach number of unity, it should be possible to calculate from a known airfoil pressure distribution the growth of the boundary layer and the position of the terminal shock wave.

In order to do this, it is necessary to relate the rise in pressure along the surface to the deflection in the stream direction. This may be accomplished by assuming that pressure changes take place on Mach lines, as was done by Ackeret and Busemann (reference 4, pp. 235-236), or by assuming that the changes occur along shock waves of infinitesimal magnitude. The latter approach will be adopted here. Consider now the uniform, two-dimensional flow of a compressible fluid at a Mach number $M_1 > 1$. If this flow is deflected through an angle θ as shown in figure 5, which is less than a certain prescribed value, theory indicates that a linear shock wave will be formed in the fluid and that the shock will pass through the point at which the deflection occurs. Moreover, classical theory has shown that it is possible to relate conditions on both sides of the shock wave by means of the equations which follow from the assumption that mass, momentum, and energy are conserved in passing through the shock (reference 4, p. 238). As a result of these assumptions it can be shown that

$$p_2 - p_1 = \rho_1 V_1^2 \frac{\sin\theta \sin\alpha}{\cos(\alpha-\theta)} \tag{1}$$

$$p_2 - p_1 = \rho_1 V_1^2 \left(\sin^2\alpha - \frac{1}{M_1^2}\right) \frac{2}{\gamma+1} \tag{2}$$

where

p pressure

ρ density

θ angle of deflection of stream

α angle of inclination of shock wave

V velocity

γ ratio of specific heats ($c_p/c_v = 1.4$)

$_{1,2}$ subscripts denoting conditions before and behind the shock, respectively.

As θ approaches zero, the pressure rise across the shock approaches zero and from equation (2) it follows that $\sin\alpha \cong 1/M_1$. Thus α is the Mach angle associated with the free-stream Mach number and the shock wave becomes in the limit a Mach line. In the analysis that follows, an expression will be derived relating the change in pressure with the value of θ.

By setting

$$p_2 - p_1 = \Delta p$$

$$\frac{1}{2} \rho_1 V_1^2 = q_1$$

$$\Delta p / q_1 = \Delta P$$

equations (1) and (2) may be written, respectively, in the forms

$$\Delta P = \frac{2 \tan \alpha \tan \theta}{1 + \tan \alpha \tan \theta}$$

$$\sin^2 \alpha = \frac{(\gamma+1)\ \Delta P}{4} + \frac{1}{M_1^2}$$

It is possible to eliminate α between the two preceding equations and, as a result, ΔP may be expressed as a function of $\tan \theta$ or, inversely, the angle of deflection may be expressed in terms of the pressure change. Solving for $\tan \theta$ gives

$$\tan \theta = \frac{\frac{1}{2} \sqrt{M_1^2 - 1}\ \Delta P}{\left(1 - \frac{\Delta P}{2}\right)} \sqrt{\frac{1 - \frac{\gamma+1}{4} \frac{M_1^2\ \Delta P}{M_1^2 - 1}}{1 - \frac{\gamma+1}{4} \frac{M_1^2\ \Delta P}{M_1^2 - 1} + \frac{\gamma+1}{4} \frac{M_1^4\ \Delta P}{M_1^2 - 1}}} \qquad (3)$$

If θ is small, it follows that the pressure increment is small and, as a consequence, the right-hand member of equation (3) can be expanded in series form. Neglecting powers of ΔP above the second yields the following

$$\tan\,\theta = \Delta P\,\frac{\sqrt{M_1^2-1}}{2} - (\Delta P)^2\,\frac{\sqrt{M_1^2-1}}{4}\left[\frac{(\gamma+1)M_1^4}{4(M_1^2-1)} - 1\right] \tag{4}$$

If, in turn, ΔP is expressed in terms of $\tan\,\theta$, and powers of $\tan\,\theta$ above the second are omitted, the result is

$$\Delta p = \rho\,V_1^2\left\{\frac{\tan\,\theta}{\sqrt{M_1^2-1}} + \frac{\tan^2\theta}{M_1^2-1}\left[\frac{(\gamma+1)\,M_1^4}{4(M_1^2-1)} - 1\right]\right\} \tag{5}$$

It is thus possible, under the assumptions of thin-airfoil theory, to treat pressure variation in a supersonic stream either as a limiting case of attenuated shock waves or by means of the previously known approach which assumes the pressure changes take place along Mach lines.

If the relation between pressure-coefficient change and angle-of-stream deflection given in equation (4) is used, it is then possible to calculate the growth of the boundary layer and the change in local Mach number by a step-by-step process.

DISCUSSION AND CONCLUDING REMARKS

The theoretical analysis was applied to predict the boundary-layer growth and the formation of shock corresponding to some measured pressure distributions. A comparison of several calculated flows with the corresponding observed flows is shown in figures 4, 6, 7, and 8. (The flows of fig. 4, as noted previously, correspond to the pressure distribution of fig. 3.) In these calculations the Mach lines corresponding with the measured surface pressures have been arbitrarily extended linearly and are terminated at the extended Mach line of unity. Such a diagram clearly cannot conform with reality at points removed from the surface since it fails to allow for the proper pressure variation normal to the airfoil. In particular, the unity Mach line, although possible close to the surface, could not extend into the flow field to intersect the other Mach lines as shown. In spite of these limitations, it is seen that, in figures 4 and 6, the terminal shock is at a constant chordwise position along the span and the calculated and experimental flows are in good agreement, indicating that the previously discussed mechanism for the interaction between shock and boundary layer is substantially correct. The schlieren photographs of figures 7 and 8 suggest that the flow varies spanwise. In spite of this, reasonable agreement is obtained. In addition, the figures show that near the surface the actual location of the shock wave is very close to the calculated position for a Mach number of unity which confirms Donaldson's hypothesis.

With a clean surface it can be assumed that, at the relatively low Reynolds number of the test, the flow remained laminar up to the shock. Although no measurements were made in the present investigation to support this assumption, a similar condition was found to exist at much higher Reynolds numbers in an investigation in the Ames 16-foot high-speed wind tunnel of a large chord NACA low-drag-type airfoil.

With the addition of carborundum to the airfoil of the present investigation, transition from laminar to turbulent flow probably moves forward enough so that a turbulent boundary layer exists over the region of interest. It is well known that while the laminar layer is easily thickened by an adverse pressuregradient, the turbulent layer is more resistant to such a gradient. It is then to be expected that the thickening of a laminar boundary layer due to the pressure rise across a given shock wave will be greater than for a turbulent boundary layer. If it is assumed that the terminal shock wave occurs at a Mach number near unity, the previous analysis has shown that the angular deflection of the stream at the outside of the boundary layer has a fixed value. Hence the marked thickening due to the terminal shock must be more limited in chordwise extent for a turbulent layer than for a laminar layer. The pressure distributions of figure 1 show this to be the case.

In the cases so far considered, sufficient thickening to induce shock near a Mach number of unity has been evidenced by the pressure distributions. It cannot be concluded, however, that all shocks must take place at close to unity Mach number since the lower-surface pressures of figure 3 would not appear to support such a conclusion. As a further warning, it should not be concluded that whenever a pressure distribution of the type shown for the upper surface in figure 3 is obtained the shock necessarily takes place near unity Mach number, unless the pressures are both instantaneous and persisting. Usually pressures are measured on liquid manometers possessing slow response which, in consequence, can indicate for this latter type of pressure distribution either a softened shock, as previously discussed, or a fluctuating-shock flow which in other instances has been observed.

Under the assumption that shock occurs at about the chordwise station where the local Mach number has dropped to unity, the pressure distributions of figure 1 indicate the following: When the boundary-layer flow is changed from laminar to turbulent, by means of the strip of coarse carborundum, the location of the shock wave is moved forward and the boundary-layer separation after shock appears complete.

As will be seen in figure 2, the growth of the boundary layer ahead of the terminal shock wave is usually one wherein the thickening of the boundary layer is at first slow and increases as the terminal shock is approached, as shown in figure 9(a). The pressure distribution is then of the type shown in figure 9(b). The Mach lines, which are but shock waves of weak strength, are generated at the boundary-layer surface and would form an oblique shock of greater strength if they coalesced to form an envelope. However, such coalescence has not space enough to occur ahead of the terminal shock in the usual case, and hence, the oblique shock would not be evident.

If, on the other hand, the boundary layer were laminar and separated to form a wedge of constant angle, as shown in figure 10(a), the envelope of Mach lines would form an oblique shock which would disappear at the terminal shock and the pressure distribution would be of the type shown in figure 10(b). That such types do actually occur is seen in figure 2 for some of the lower-surface shocks.

It is important to note that, with the addition of the carborundum, the reduction of lift was sizable but that the relative change in drag was slight. In figure 11 is shown the pressure drag coefficient (which at speeds well in excess of the critical is practically the total drag coefficient) for the airfoil whose pressure distributions are shown in figure 1. It appears that, whether or not the energy loss in the boundary layer is increased with a consequent reduction in energy loss in the shock wave, the net effect on the total drag is practically the same. It should be noted, however, that in the cases considered, a softened shock is apparently always attained and the conclusion may not be warranted for other types of shocks. Finally, it should be observed that although the drag is relatively insensitive to changes in the type of boundary-layer flow these changes produce important shifts in the location of the shock wave and in the lift. Therefore, increase of Reynolds number, in its effect on thinning the boundary layer relative to the airfoil dimensions may have an important effect at high Mach numbers.

Ames Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Moffett Field, Calif.

H. Julian Allen,
Aeronautical Engineer.

Max. A. Heaslet,
Physicist.

Gerald E. Nitzberg,
Aeronautical Engineer.

Approved:

Donald H. Wood,
Aeronautical Engineer.

REFERENCES

1. Pinkerton, Robert M.: Calculated and Measured Pressure Distributions over the Midspan Section of the NACA 4412 Airfoil. NACA Rep. No. 563, 1936.

2. Nitzberg, Gerald E., and Sluder, Loma E.: An Empirical Theory of Pressure Distributions over Airfoils at Supercritical Mach Numbers and Moderate Angles of Attack. NACA Proposed ACR, Aug. 29, 1945.

3. Donaldson, Coleman duP.: Effects of Interaction between Normal Shock and Boundary Layer. NACA CB No. 4A27, 1944.

4. Taylor, G.I., and Maccoll, J.W.: The Mechanics of Compressible Fluids. Vol. III, div. H, ch. 4 of Aerodynamic Theory, W.F. Durand, ed., Julius Springer (Berlin), 1935.

[redacted]

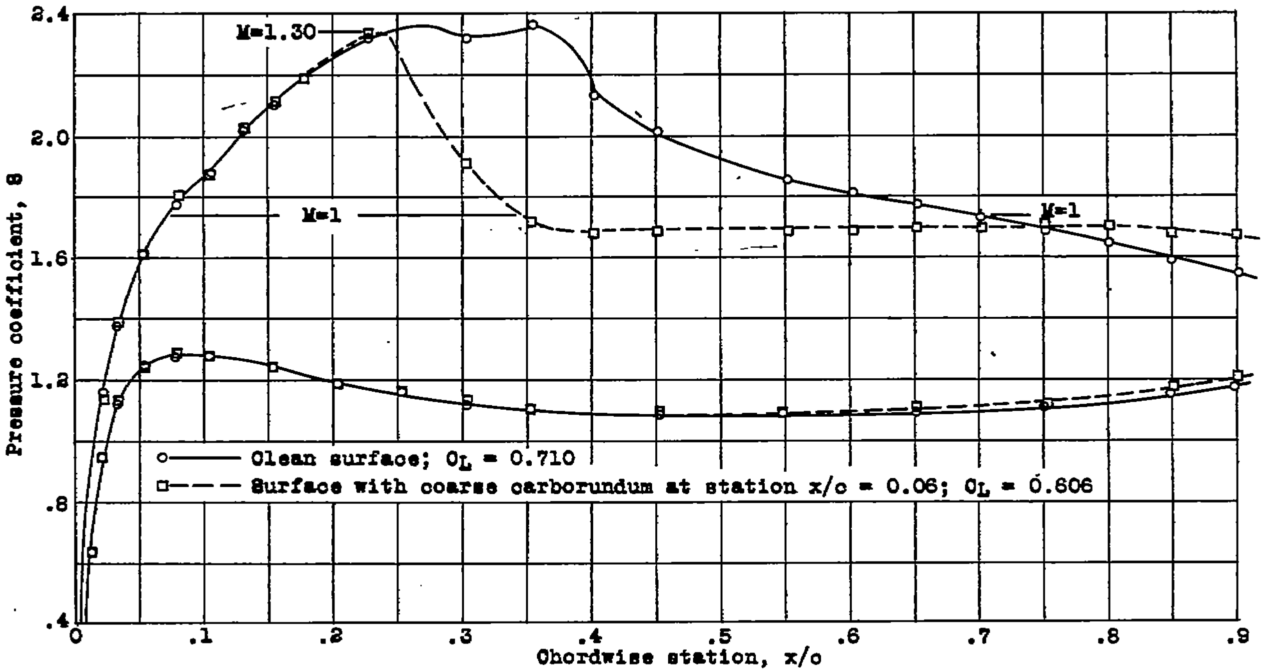


Figure 1.- Effect of surface condition on the distribution of pressures over an NACA 4412 airfoil at 4° angle of attack, Mach number 0.75, and Reynolds number 1.5 millions.

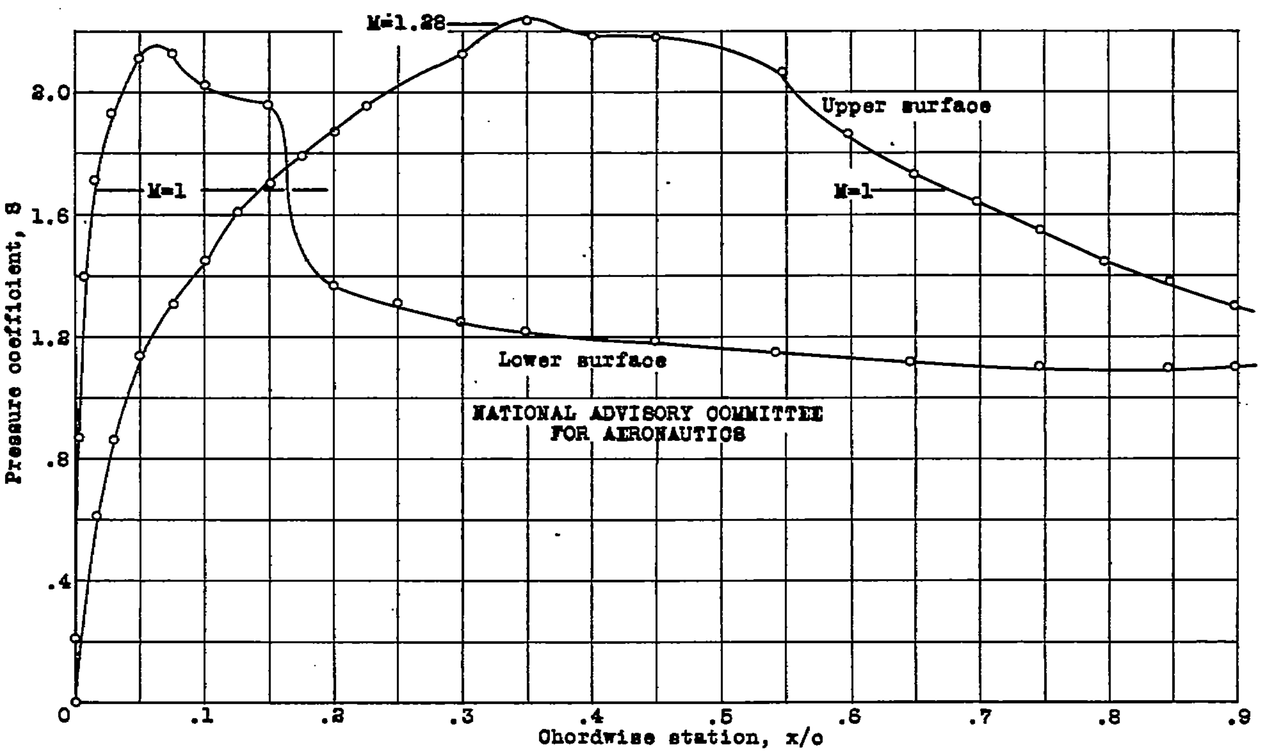
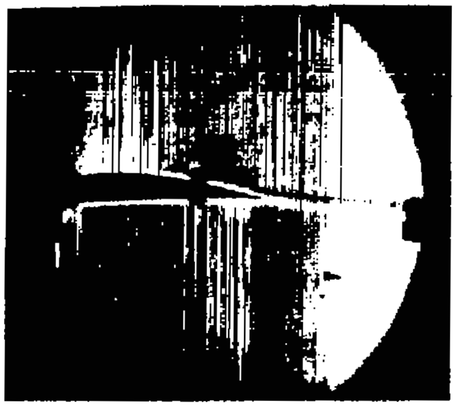
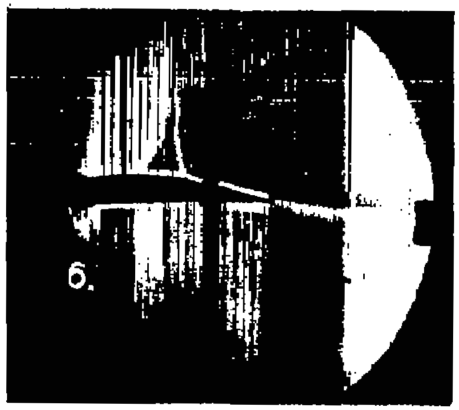


Figure 3.- Distribution of pressures over an NACA 4412 airfoil section at 0° angle of attack, Mach number 0.772, and Reynolds number 1.5 millions.

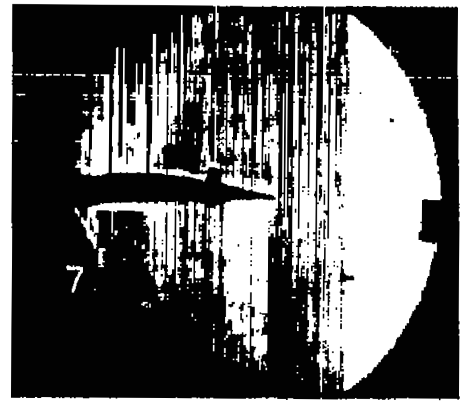
[redacted]



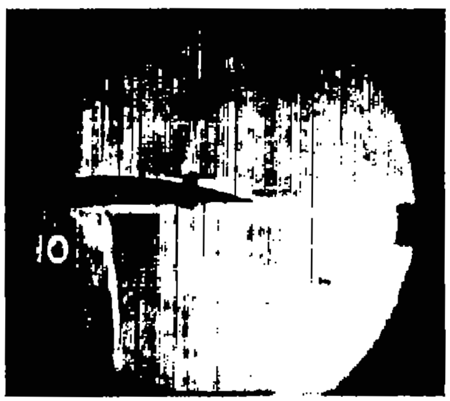
$\alpha = -2^{\circ}$; M = 0.700.



$\alpha = -2^{\circ}$; M = 0.750.

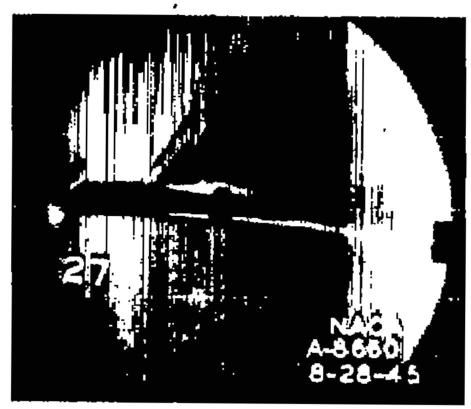


$\alpha = -2^{\circ}$; M = 0.775.



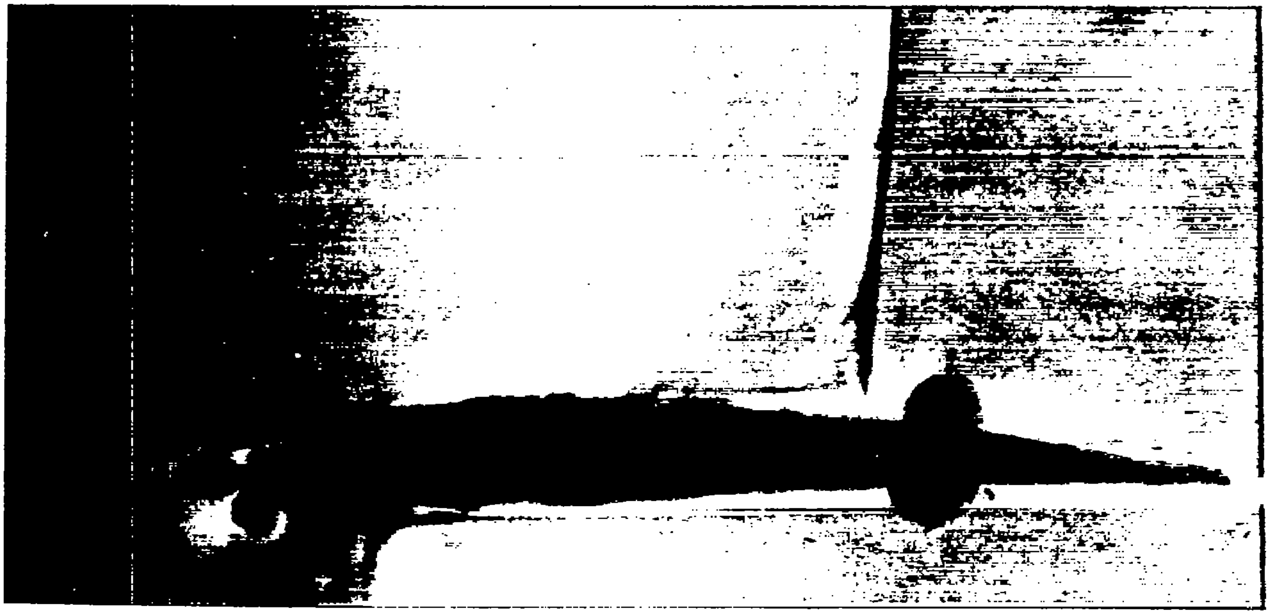
$\alpha = -2^{\circ}$; M = 0.800.

$\alpha = 2^{\circ}$; M = 0.725.

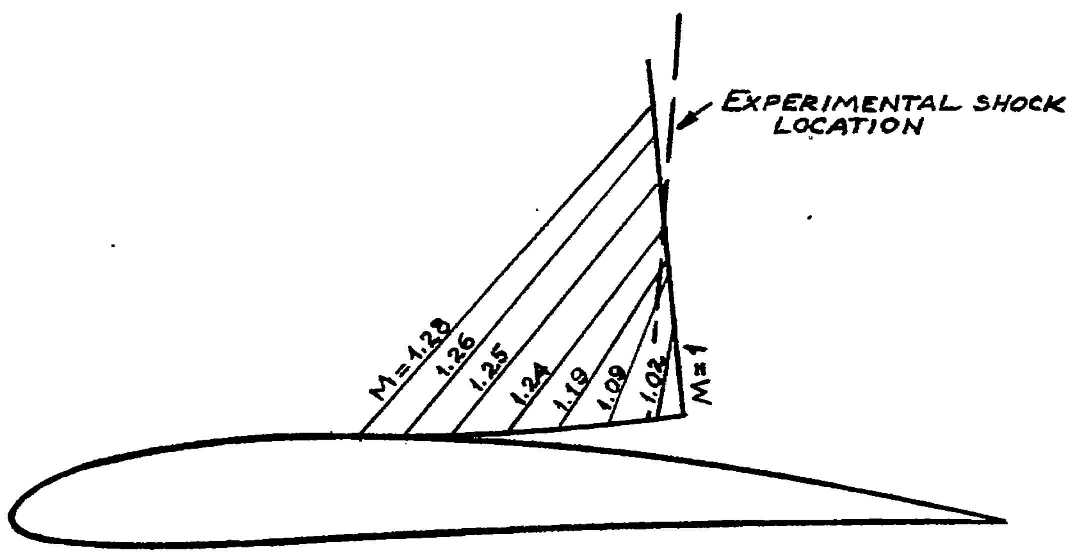


$\alpha = 2^{\circ}$; M = 0.775.

Figure 2.- Representative schlieren photographs of the flow about an NACA 4412 airfoil at several Mach numbers and angles of attack.



(a) EXPERIMENTAL FLOW PATTERN.



(b) CALCULATED FLOW PATTERN.

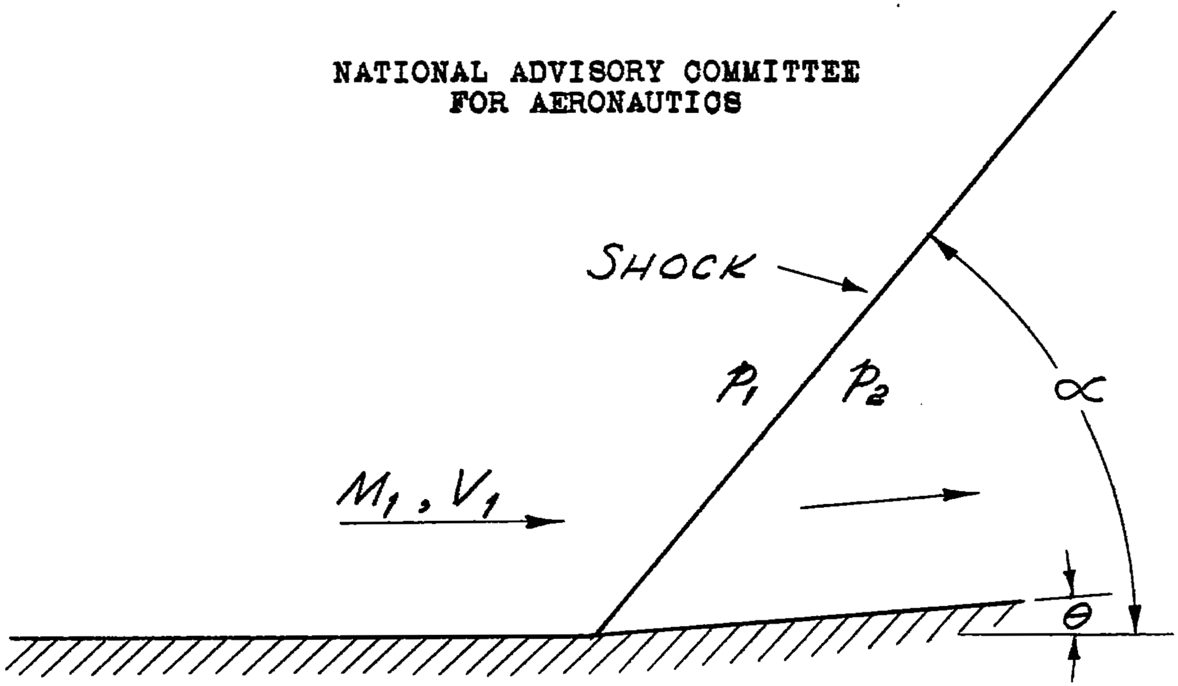
NACA
A-8661
8-29-45

FIGURE 4.- CALCULATED AND EXPERIMENTAL FLOW ABOUT AN NACA 4412 AIRFOIL SECTION AT 0° ANGLE OF ATTACK AND MACH NUMBER 0.772.

[redacted]

[redacted]

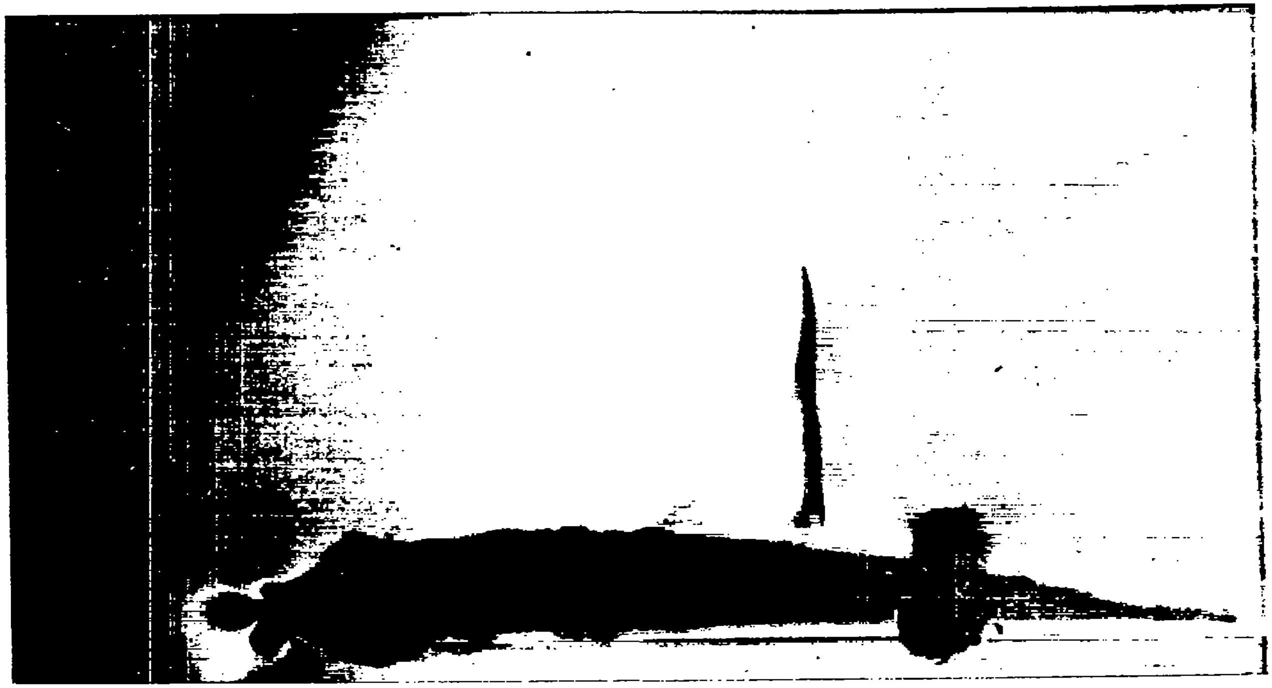
NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS



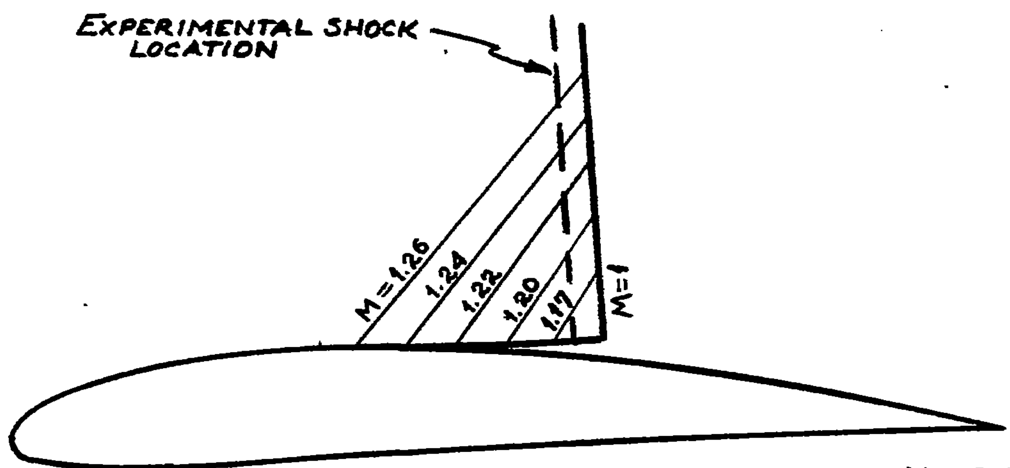
[redacted]

FIGURE 5.- SHOCK WAVE PRODUCED BY ANGULAR DEFLECTION OF UNIFORM SUPERSONIC STREAM.

[redacted]



(a) EXPERIMENTAL FLOW PATTERN.



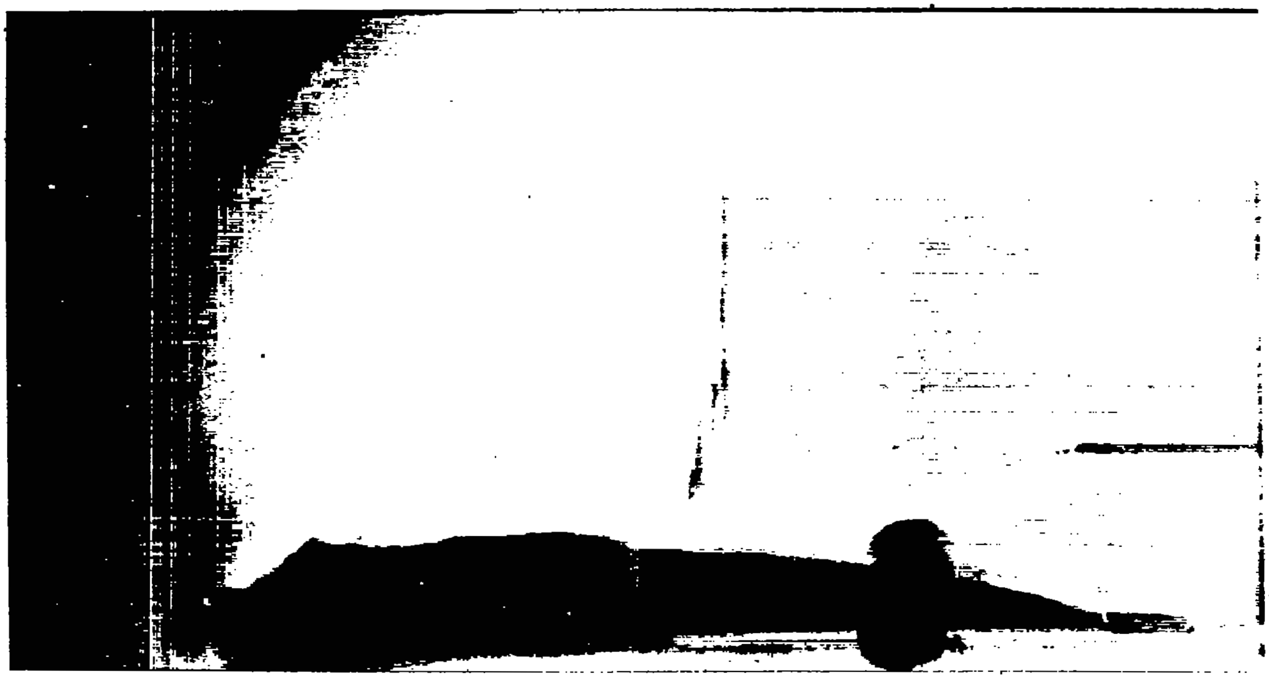
(b) CALCULATED FLOW PATTERN.

NACA
A-8662
8-29-45

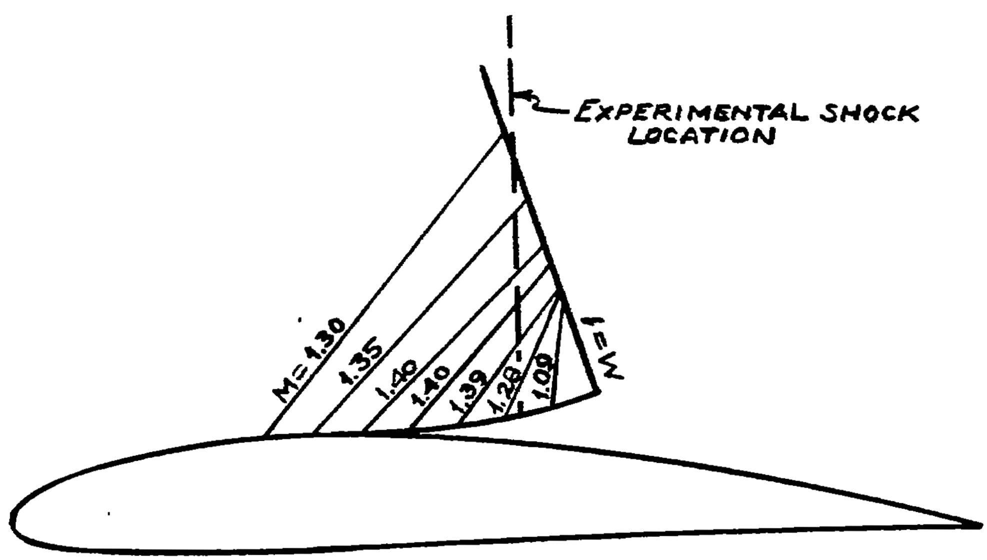
[redacted]

FIGURE 6.- CALCULATED AND EXPERIMENTAL FLOW ABOUT AN NACA 4412 AIRFOIL SECTION AT 0° ANGLE OF ATTACK AND MACH NUMBER 0.722.

[redacted]



(a) EXPERIMENTAL FLOW PATTERN.



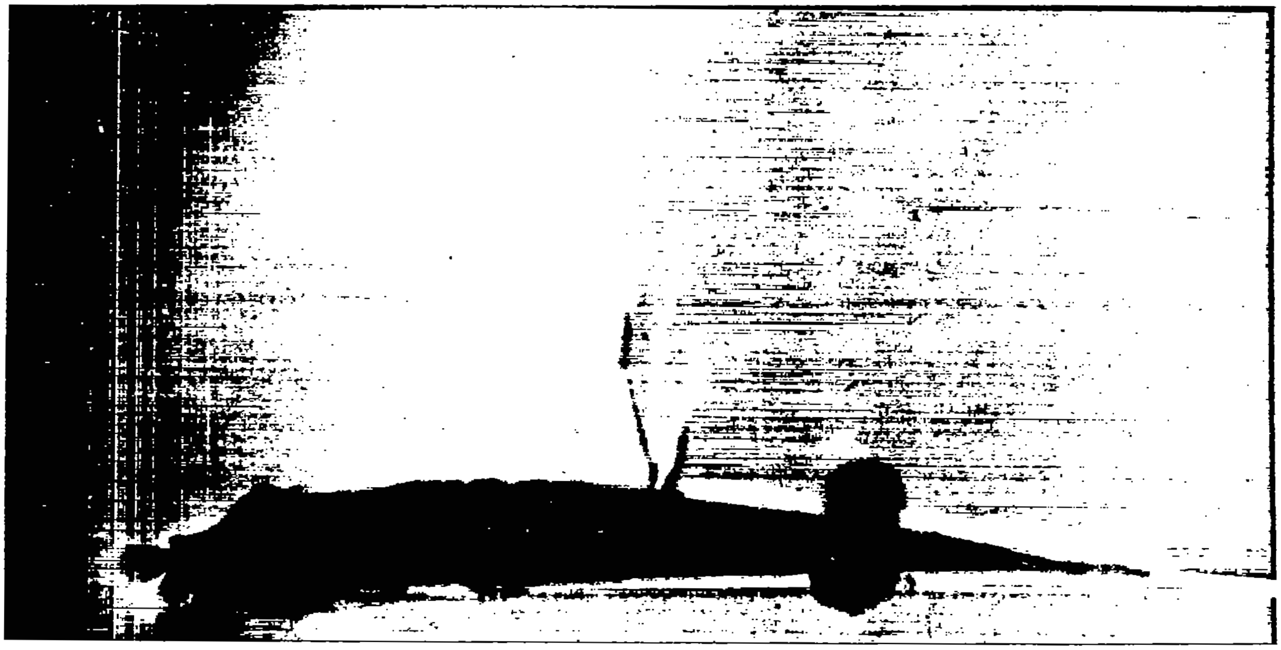
(b) CALCULATED FLOW PATTERN.

NACA
A-8663
8-29-45

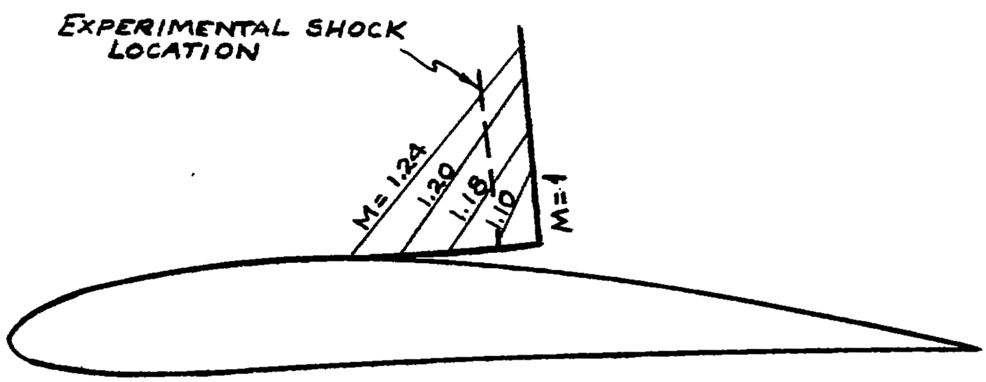
[redacted]

FIGURE 7.- CALCULATED AND EXPERIMENTAL FLOW ABOUT AN NACA 4412 AIRFOIL SECTION AT 4° ANGLE OF ATTACK AND MACH NUMBER 0.750.

[redacted]



(a) EXPERIMENTAL FLOW PATTERN.



(b) CALCULATED FLOW PATTERN.

NACA
A-8664
8-29-45

[redacted]

FIGURE 8.- CALCULATED AND EXPERIMENTAL FLOW ABOUT AN NACA 4412 AIRFOIL SECTION AT 0° ANGLE OF ATTACK AND MACH NUMBER 0.700.

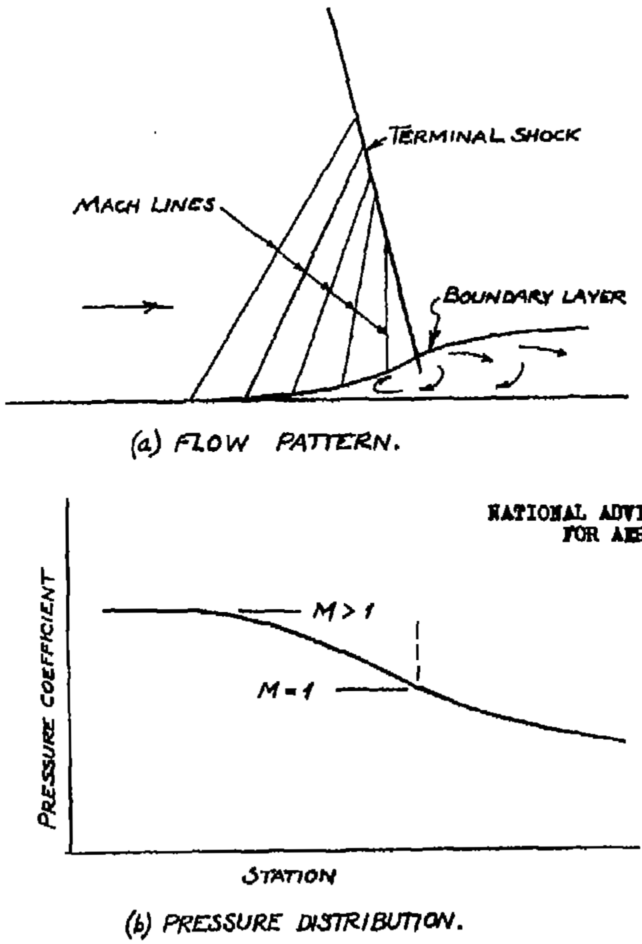


FIGURE 9.- INTERACTION OF SHOCK WAVE AND BOUNDARY LAYER FOR SINGLE-SHOCK FLOW.

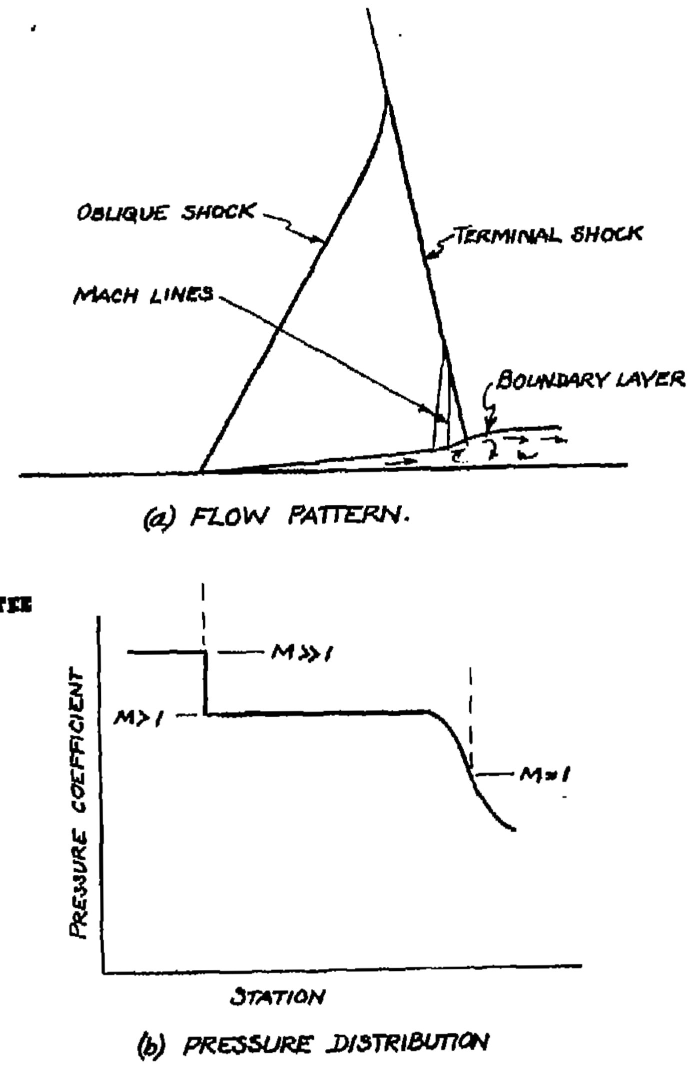


FIGURE 10.- INTERACTION OF SHOCK WAVE AND BOUNDARY LAYER FOR DOUBLE-SHOCK FLOW.

[redacted]

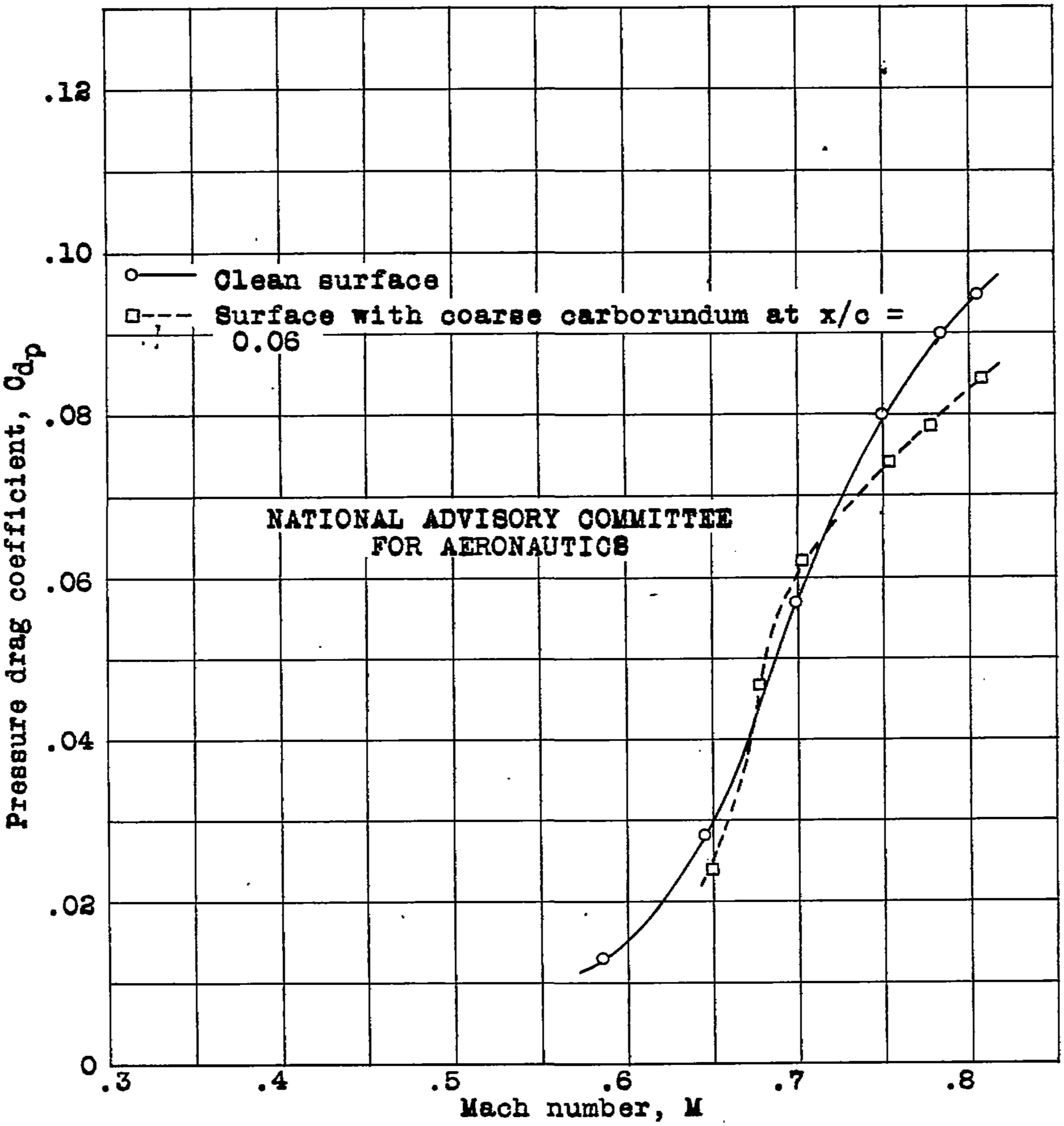


Figure 11.- Effect of surface condition on the pressure drag of an NACA 4412 airfoil at 4° angle of attack.

[redacted]